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**APPLICATION OF COMPOSITES TO THE SELECTIVE  
REINFORCEMENT OF METALLIC AEROSPACE STRUCTURES**

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# APPLICATION OF COMPOSITES TO THE SELECTIVE REINFORCEMENT OF METALLIC AEROSPACE STRUCTURES

by

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## SUMMARY

The use of composite materials to selectively reinforce metallic structures provides a low-cost way to reduce weight and a means of minimizing the risks usually associated with the introduction of new materials. In this paper, an overview is presented of the NASA Langley Research Center programs to identify the advantages and to develop the potential of the selective reinforcement approach to the use of composites. These programs have shown that selective reinforcement provides excellent strength and stiffness improvements to metallic structures. Significant weight savings can be obtained in a cost effective manner. Flight service programs which have been initiated to validate further the merits of selective reinforcement are described.

## INTRODUCTION

The development of composite materials for application to aerospace structures is currently being emphasized in the United States. Increased emphasis is justified due to the growing technological base that has been developed over the past 10 years and the great potential payoff that current systems studies are indicating. The subject of most of this attention is the so-called "all-composite" structural component.

Another concept which has received less attention is that of selective reinforcement of metallic structures with filamentary composite materials. Many consider the selective reinforcement concept as an evolutionary approach to the all-composite structure. Indeed, selective reinforcement may well be the initial approach that airframe manufacturers will employ to gain the benefits of composites in the near future.

The use of the selective reinforcement concept may be justified for several reasons. Because smaller amounts of the new composite materials are required and because design criteria tend to be conservative, the risk is relatively small. Furthermore, the use of smaller amounts of composite materials can be cost effective in this era when composite materials are still expensive and the increased initial cost of all-composite structural components is more than some customers are willing to pay. An advantage also exists in manufacturing because existing metal forming and machining technology, production experience, and equipment can be used for a major portion of the structure. Laminate construction is reduced to its simplest form, particularly when the fibers are all aligned in one direction to take maximum advantage of their strength and stiffness. Lastly, attachment problems that are encountered with composite materials can be minimized in the selective reinforcement approach by making transitions to all-metal sections for joining.

Several years ago, the NASA Langley Research Center recognized the many advantages and possibilities inherent in the selective reinforcement approach to the use of composites and initiated a technology development program. The purpose of this paper is to provide a brief overview of the Langley program which has shown that significant and cost effective improvements in load-carrying capability can be achieved. (Details are given in References [1] - [14] produced by, or for, NASA Langley Research Center.)

The advantages of the selective reinforcement concept and some of the problems encountered in its application will be discussed. The results of various engineering studies will be presented showing weight savings, costs, and comparisons of different concepts. Finally, flight service programs which are necessary to develop broader acceptance of, and confidence in, composites will be described.

## GENERAL CONSIDERATIONS

### Reinforced Tubes

Some of the early studies of reinforcing structural elements were conducted using tubular compression specimens [1 and 2]. The compressively loaded tube was selected because of convenience and the availability of well-developed analyses. Furthermore, a simple and effective method had been developed for applying the reinforcing composite [3]. Briefly, this method consists of laying the composite on the metal

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tube as desired, enclosing the assembly in a heat-shrinkable plastic sleeve, heating to compact the plies, and then curing.

Typical results from Reference [2] are shown in Figure 1. The specific strength and stiffness of aluminum tubes reinforced with boron-epoxy are shown as a function of the ratio of composite volume to total volume. These results are for tubes with approximately the same mass and tested in axial compression. The effect of using composites is shown quite dramatically by this figure which indicates that for the all-composite tube (composite volume/total volume = 1), the specific strength is 10 times and the specific stiffness is over four times that of the all-aluminum tube. Even small amounts of reinforcement produce significant changes. Furthermore, the results show that composite-reinforced metal structures can be tailored to produce particular values of specific strength or stiffness.

In another study of reinforced tubes for truss structures, costs were considered as well as strength and stiffness. Results from this study [4] are shown in Figure 2. The example considered is a 1.1-meter tubular column designed to carry a compressive load of 790 kN. Costs and weights associated with three designs are shown: all titanium, 33% boron-epoxy composite and 67% titanium (by mass), and all-composite. The solid curve is based on actual costs and includes boron-epoxy prepreg material at \$485 per kg and fabrication at \$220 per kg. If boron-epoxy were available at \$220 per kg, the result would be as shown by the dashed curve.

In order to achieve the maximum weight saving (about 65%) for the actual-cost case, the total tube cost increases from \$400 to \$1000. This means that the value of each kilogram saved must be greater than the \$200 per kg that the saving costs if the all-composite tube is to be justified. On the other hand, for a column with 33% boron-epoxy reinforcement and 67% titanium, the weight can be decreased 54% with a 50% cost increase and the cost of each kilogram saved is \$80. Of course, the cost-weight relationship is sensitive to material and fabrication costs.

The effect of boron-epoxy costs is shown by the dashed curve. As can be seen, for \$220 per kilogram boron-epoxy prepreg, the maximum weight saving provided by the all-composite tube can be achieved for approximately the same cost per kilogram of saving as the reinforced tube on the solid curve. However, in this case (the dashed curve) the composite-reinforced tube provides a 54% weight saving with less than 10% cost increase.

#### Reinforced Stiffened Panels

Stiffened panels, such as shown in Figures 3 and 4, have also been studied to determine the effects of reinforcement. The hat stiffener shown in Figure 4 is generally favored over the Z-stiffener of Figure 3. Z-stiffeners with boron-epoxy composite on the outstanding flanges do not appear to be efficient structural elements [5]. This results because the composite with all fibers parallel to the stiffener has little twisting stiffness and causes a limiting mode of flange buckling.

An example of reinforced hat stiffeners is shown by Figure 5. In this case, boron-aluminum composites with uniaxial fibers have been brazed to the stiffeners of a titanium panel [6]. Experimental and calculated maximum compressive strength-weight ratios of reinforced and unreinforced titanium panels are shown as a function of temperature. As a result of the addition of boron-aluminum, the reinforced panels weigh approximately 9% more than the unreinforced titanium panels. The improvement in strength gained by the use of composites is about 30% for the range of temperatures indicated. Although there is good agreement between calculated and experimental results for the unreinforced panel, the calculated results are high for the reinforced panel. This difference is attributed to the fact that the initial elastic modulus of the bilinear stress-strain curve of boron-aluminum was used in the calculation because the secondary modulus was not known.

In yet another study [7], compression-critical composite-reinforced stiffened panels designed to the same constraints and loads as a current subsonic commercial transport and a proposed supersonic transport were investigated. Some of the results are shown in Figure 6. The relative weights of various boron-epoxy reinforced panels are shown plotted against load intensity. For a load intensity of approximately 1.4 MN/m, which is representative of the subsonic aircraft, the reinforced panels with boron-epoxy composite located as shown by the cross-hatched regions provide a weight saving of approximately 30% when compared to the all-aluminum structure designed for the same buckling load. For the supersonic case with a load intensity of 3.2 MN/m, a weight saving of 15% over the all-titanium panel is indicated for the reinforced titanium panel.

In the tests which produced these data, skin buckling generally took place before the panels failed. This introduced large peel stresses on the stiffener-skin bond, causing it to separate and initiate panel failure.

Another selective-reinforcement concept for panels [8] is shown in Figure 7. The photograph in the lower left of the figure shows the type of construction investigated. Aluminum sections are extruded with axial holes; these holes are then filled with epoxy-coated boron fibers. Next, additional resin is infiltrated and cured in place to form a composite section — much like the RAE "pultrusion" process. Compression panel design studies indicate that configurations such as that shown may be 25% lighter than optimally designed all-metal panels.

The photographs on the right of the figure show a short panel that has been tested in compression. As indicated, the experimental and calculated crippling strengths are in excellent agreement. A panel-instability test was also conducted on a larger panel. However, that panel failed prematurely at 64% of ultimate design load. The premature failure may have been initiated by debonding of the boron-epoxy inserts.

### Residual Stresses

One of the major problems associated with composite reinforced metal structures results from the different thermal expansion characteristics of the constituents. When the composite, which has a lower coefficient of expansion, is joined to the metal at one temperature and is cooled to another temperature, a state of thermally induced stress exists.

Elementary residual stresses are given in Figure 8 for an aluminum member uniaxially reinforced by boron- or graphite-epoxy as shown in the sketch. If all of the aluminum and the composite are assumed to behave like two rods bonded together at 450 K and bending is prevented, the residual stresses that develop during cooling are as shown in the figure, provided that the rods were stress-free at the bonding temperature. The shaded bands give the range of stresses produced by various composites. The lowest values of tensile and compressive stresses result from medium-strength graphite; the intermediate values, from boron or high-strength graphite; and the highest values from high-modulus graphite. The most significant result is that the aluminum contains high residual tensile stresses that could lead to early fatigue failure in the presence of repeated loads. Furthermore, the reinforced metal component will bend to relieve stresses, producing warpage that could present difficulty during assembly.

Relief of residual stresses and minimization of warpage has been the subject of much research (for example, [7] and [9]). In a given combination of composite and metal, the parameters that can be varied are the cure temperature of the adhesive, the degree of expansion restraint imposed on the metal, and the degree of preextension imposed on the composite.

Some of the techniques investigated are illustrated in Figure 9. The unrestrained autoclave cure at elevated temperatures results in the situation shown in Figure 8. If the adhesive can be cured at a lower temperature, stresses at room temperature will be reduced. However, low-temperature-curing adhesives that possess both adequate strength and environmental resistance are not presently available.

During this discussion of cure temperatures, it should be pointed out that the cure temperature is not necessarily the stress-free temperature [7]. Some adhesive cure cycles require holding at an intermediate temperature. Depending upon the adhesive and the temperature level, sufficient cross-polymerization may take place to bond the two materials and resist further differential expansion at the final cure temperature. When this happens, the stress-free temperature is the intermediate temperature.

The second method, shown in Figure 9, is the restrained autoclave technique. The sketch shows the metal structure being restrained by stops fastened to the tool which is usually steel. Of course, the steel tool is heated in the autoclave and the restraining force is relieved somewhat by the expansion of the steel. The metal component may alternatively be mechanically fastened to the tool. Furthermore, it has been suggested that the composite be stretched during autoclaving while the metal is being restrained or compressed. Yet another alternative is to mechanically fasten the composite and metal structure before autoclaving to cause the stress-free state to be at room temperature. Some of these techniques will not produce completely satisfactory results.

The last concept shown is called the "cool tool" method [10]. In this method, heating is produced by a heating blanket that is placed between the metal component and a layer of insulation that keeps the tool relatively cool, thus maximizing the restraint. Pressure is applied to the composite by air bags. If the restraining stops fit snugly against the metal component before heating, this method produces a small residual compressive stress in the metal at room temperature. Preloading the stops before heating will produce a greater residual compressive stress in the metal. Thus, the residual stresses can be selected for a particular operating temperature.

### Load Transfer Joints

Another significant problem area associated with composite-reinforced metal designs is the transfer of load from the composite to the metal component. The basic criterion for designing the load transfer region is to provide equal stiffness load paths to each laminate and fiber. In practice, difficulties are usually encountered in meeting this criterion.

Commonly used concepts for load transfer joints are indicated in Figure 10. The first shown is the stepped concept that has a titanium end fitting. The metal fitting is fabricated with small steps sized to accommodate one or more plies of the composite. The length of the steps is of the order of 1 cm. The design of the load transition fitting (material and step configuration) can greatly influence the failure modes for static loading, cyclic loading, and creep.

An alternate approach consists of interleaving metal shims between the stepped-down plies of the composite and using mechanical fasteners to transfer the loads to the metal component. As noted, the

shims are used to increase the bearing strength at the joint. However, another advantage of this concept is its ability to resist composite peeling failures which can occur in certain buckling modes.

The last concept shown, called "run-out," has been used for designs that were stiffness critical and did not involve high loading of the metal component. Again, a series of steps in the composite are utilized to avoid an abrupt change in stiffness. In one case [11] where the composite was boron-epoxy, the transition was made more gradual by the insertion of two layers of unidirectional glass/epoxy, each about 5 cm long, between the composite and the metal component at the ends of the tapered joints. This modification reduced the peak shear stress in the adhesive by approximately 50%.

An example of the performance of a composite-reinforced metal panel containing a stepped joint is shown in Figure 11 [12]. The composite-reinforced metal specimen was subjected to fully reversed loading ( $R = -1$ ) to determine fatigue life. The composite was boron-epoxy unidirectional material with a titanium load transfer fitting and the metal was 7075-T6 aluminum alloy. The solid curve pertains to the composite-reinforced metal specimen. The dashed curves apply to equal mass 7075-T6 aluminum alloy specimens with a  $K_T$  (stress concentration factor) of 1.5 (representative of bonded construction) and 4.0 (representative of riveted construction). The advantage of the composite-reinforced metal component is clear at the lower values of load cycles. At the higher values of load cycles (greater than  $10^5$ ), the composite-reinforced metal component results approach that of the aluminum with  $K_T = 1.5$ . However, the fatigue life for the composite-reinforced metal specimens is still considerably greater than that of the aluminum with  $K_T = 4.0$ .

The modes of failure observed in these tests were influenced by the transfer fitting design and the composite matrix strength. The failures included progressive debonding of the composite and cracking at one of the steps in the metal end fittings. Further research to provide ways to improve the fatigue life of the composite-to-metal load transition joints is needed before the performance gain indicated in Figure 11 can be achieved in aircraft structural components.

## APPLICATIONS STUDIES

### Aircraft Fuselage Panels

In order to extend the development of various reinforced metal concepts and to provide data on performance, three types of panels that would simulate the design features of full-scale aircraft components were selected for study [13]. The concepts selected had conventional metal-construction counterparts for which design and experimental data existed. Thus, straightforward comparisons could be made between the reinforced metal and all-metal components.

The composite-reinforced metal panel, shown in Figure 12, was designed to contain penetration damage under internal pressure loading. The design, shown by the sketches, is an aluminum honeycomb-core and skin panel stabilized by aluminum frames. The panel was reinforced with boron-epoxy composite under the aluminum skins as shown by the bottom right sketch. The skins and honeycomb core of the panel were designed to meet the maximum side- and vertical-bending load conditions. The boron-epoxy was applied with the fibers in the circumferential direction to increase the panel strength to the point that it would meet the hoop tension and penetration damage containment requirements. The panel was mounted in a fixture that provided realistic loading, pressurized, and penetrated by a 30-cm steel blade at a frame location to check damage containment. The penetration produced a catastrophic failure. Subsequent analysis indicated that residual thermal stresses, which add to the pressure stresses, had not been included in the panel design. However, the weight saving potential was estimated to be about 20% with proper consideration of the residual stresses.

Another panel was designed for application on the pressurized, lower aft fuselage of a supersonic transport. This design, shown in Figure 13, is a titanium skin-stringer panel attached to channel frames. The hat-section stiffeners were reinforced with unidirectional boron-epoxy as shown. The panel was required to carry the loads indicated in the figure and was compression critical. However, elastic skin buckling was allowed.

Two aft-fuselage panels were fabricated and tested. One of the panels failed prematurely as a result of faulty bonding of the boron-epoxy reinforcement. However, the second panel failed at 111% of the design ultimate load after being cycled 100 times to limit load. Furthermore, this was accomplished with a 34% weight saving over the corresponding all-titanium panel.

The third type of panel was designed, fabricated, and tested to determine the effectiveness of the selective reinforcement concept when applied to cutouts in shear critical structure — the window belt panel of a commercial transport. This panel, shown in Figure 14, experiences combined shear, hoop tension, and side bending tension. Four load cases were considered and the principal loads are given in the table of Figure 15.

Details of the panel design are given in the sketches in Figure 14. Briefly, the panel consisted of an aluminum honeycomb-core sandwich panel with titanium face sheets, boron-epoxy composite reinforcement between the face sheets, and a reduced depth core around the window. The boron-epoxy, which terminated on stepped fittings around the windows, is arrayed in the form of longitudinal and circumferential

doublers. Within the doublers, plies were oriented at  $0^\circ$ ,  $\pm 45^\circ$ , and  $90^\circ$  to provide a multidirectional load-carrying capability.

The panel was tested in pure shear by use of a picture-frame fixture. The most critical load case is not the pure shear, or vertical bending, load but is the combined load of internal pressure and vertical bending. When this was taken into consideration, the failure shear load was calculated to be 120% of the design ultimate (Fig. 15).

The weight of the composite-reinforced window belt panel was calculated to be 25% less than the equivalent metal component. However, the actual weight saving of the composite component was only 12%. This was due to more and heavier adhesives being used and to allowable deviations from the nominal dimensions of the metal parts.

#### Fuselage Section

Following the investigation of fuselage panels for specific locations, an engineering study [14] was made of an entire fuselage section. The section selected and the scope of the study are shown in Figure 16. All of the enumerated components of the 4.6-m-long section were investigated for varying degrees of composite utilization.

The study extrapolated the results of the fuselage section investigation to include the entire fuselage, then went on to include the cascading weight saving for the entire aircraft, and finally considered a simple economic study of an entire fleet to show the benefits of composites. Although the study provided much detail, only a brief description is presented in this paper and that concentrates largely on a comparison of the various degrees of composite utilization.

General design criteria were established to insure that the ultimate strength, fatigue resistance, and damage containment would equal or exceed those of the existing aircraft. A specific requirement was that the reinforced metal concepts be designed so the metal alone would carry limit load and the composite reinforcing would add the margin needed to sustain ultimate load. This criterion is conservative but does provide design confidence as well as adequate safety. Undoubtedly, more appropriate criteria will be developed when the total capability of composites is better defined and understood.

Although boron-epoxy, graphite-epoxy, and PRD-49-epoxy were considered for reinforcement, the final choice was high-strength graphite-epoxy. The selected material was used in three general concepts: unidirectional reinforcement, uni- and multidirectional reinforcement, and all-composite.

The results of the study projection to the entire fuselage are shown in Figure 17. Weights, weight saving, and costs are shown for employing each of the three concepts in the shaded portion of the fuselage. The weight savings range from 7.8% for the unidirectional reinforcement to 14.0% for the all-composite design. It should be noted that for the fuselage section investigated the weight savings ranged from 22 to 28%. When the entire fuselage was considered, the weight savings were not as high because substantial areas of the fuselage (door assemblies, wheel wells, windows, bulkheads, etc.) were not redesigned with composites. In addition, Concept 1 was restricted to the use of a metal skin which for large areas of the fuselage could not be reduced in thickness to gages below that of the baseline design and still meet damage containment criteria.

The economic portion of the study showed that with graphite cost at \$ 130 per kilogram and with the present value of future cost and insurance savings based on a 15% annual discount rate, only the unidirectional reinforcement concept is cost effective. The all-composite concept becomes cost effective if graphite costs are reduced to \$ 77 per kilogram.

#### Space Shuttle Components

In addition to investigating the benefits of selective reinforcement when applied to aircraft, similar studies have been conducted for structural components of the Space Shuttle [4] and [8]. The selective reinforcement concept was chosen as it was judged to have the best near-term payoff for shuttle components. This concept also makes weight savings possible without major structural configuration changes and, if needed, could be employed to preserve the very small shuttle payload margin.

Five structural components believed to encompass essentially all significant design problems were selected for investigation. One of the studies [4] was directed toward selective reinforcement of the thrust structure of an early booster configuration. A model of the tubular thrust structure is shown in the center of Figure 18. A reinforced titanium tube truss was designed with 75% boron-epoxy reinforcement and 25% metal by volume. Stepped titanium joint clusters were used to connect the truss members.

Subsequently, the one-third-scale test model, shown on the left, was fabricated with representative compression and tension members. The truss specimen was loaded and failed at 118% of the design ultimate load. A 24% weight saving over the full-scale all-titanium design was established.

Three other studies [8] are depicted in Figure 19. The large frame in the cargo bay has been designed with selective composite reinforcement. As indicated by the circled sketch, the frame

configuration is a titanium I-section and web. The frame caps are reinforced with layers of boron-epoxy which is protected by thin titanium skins. The thickness of the composite is tapered along the frame caps to accommodate the variable bending moments. The weight saving predicted for this design is 29% compared to an optimized all-titanium design. A one-third-scale model of half of the frame is to be built and tested to confirm analytical results and the calculated weight saving.

Shear web configurations for a beam-type thrust structure were also investigated. The selected design is shown at the right of Figure 19. The shear web consists of  $\pm 45^\circ$  plies of boron-epoxy sandwiched between two thin sheets of titanium. Aluminum stiffeners reinforced with boron-epoxy were used to stabilize the web. The weight saving of 30% over an all-metal design is also to be verified by a scaled test model.

The last component, shown at the bottom of Figure 19, is a fuselage panel designed for a light load condition. The composite-reinforced panel was designed for representative combined compression and shear loadings in the lower aft fuselage of an early orbiter design. Buckling of the panel skin was not permitted since this could impair the surface insulation which is bonded directly to the panel skin. The panel design selected has titanium skin and boron-aluminum hat-section stiffeners and is attached to titanium frames. The panel was found to be so lightly loaded that the usual selective reinforcement of metal stiffeners was ineffective. The weight of the boron-aluminum reinforced panel designed to operate at  $590^\circ\text{K}$  was 22% less than the all-aluminum baseline design operating at  $295^\circ\text{K}$ . However, due to the higher operating temperature of the composite-reinforced panel, less external insulation was required and the total weight savings over the insulation-protected all-aluminum design was estimated to be 46%. The weight of the reinforced panel is also 35% less than that of an all-titanium panel.

A summary of the shuttle components investigated is shown in Figure 20. Three of the component designs must be confirmed by testing. However, it is evident that the selective reinforcement concept can produce substantial weight savings for the shuttle orbiter.

#### FLIGHT SERVICE PROGRAMS

The results discussed thus far have been obtained from both analytical and experimental ground-test programs. Programs such as these are required to develop technology. Eventually, the need arises to demonstrate and verify the developed technology. With new materials, flight service programs are usually necessary to develop confidence and to obtain the most realistic service conditions.

One such Langley Research Center program is underway with the CH-54B helicopter shown by the sketch in Figure 21. The original airframe structure was designed for static loads due to flight and ground conditions. However, during developmental flight, certain lifting configurations produced undesirable dynamic conditions which required more vertical bending stiffness in the 6-m-long aft fuselage (or tail cone), as shown in the figure. The resulting production design required heavy top and bottom skins causing the aft fuselage weight to be 175 kg. The manufacturer, working on a joint NASA/U.S. Army program, designed and fabricated a tail cone that had thinner skins in the top and bottom [11]. These skins were sufficient to meet the static-strength requirement. Boron-epoxy strips were then bonded to the stringers, as indicated, in sufficient quantity to meet the additional requirement for vertical bending stiffness under dynamic response conditions. The reinforced design weighs 118 kg, a saving of 30%.

Flight qualification tests have been completed and the tail cone has been installed in a helicopter that is experiencing routine flight service. The composite material behavior will be monitored closely for at least the first 2 years. Thus far, approximately 100 hours of flight service has been accumulated with no evidence of problems with the composite.

Another program aimed at obtaining longtime flight service experience with composite materials in a primary structure involves the C-130 center wing box shown in Figure 22. The C-130 transport airplanes have experienced a rapid accumulation of fatigue damage in the U.S. Air Force service, and a number of them have been retrofitted with a strengthened aluminum center wing box to alleviate fatigue problems. A recent study [9] indicated that in place of the strengthened aluminum box, about 230 kg of boron-epoxy bonded to the skin and stringers of this 11-m-long box, as shown, can reduce the stress levels and thus increase the fatigue life as much as the aluminum retrofit design, but with a 13% weight saving.

The joint NASA/U.S. Air Force program currently consists of fabricating three wing boxes, one for ground testing, and two for installation in airplanes that will be flown in regular Air Force service. The advanced development phase has recently been completed wherein several large components were fabricated and tested as shown in Figure 23. Compression panels, tension panels, and composite-to-metal joints were fabricated and subjected to static and cyclic loading to determine strength and fatigue life. The test results, which are summarized in Figure 24, indicate that the composite-reinforced metal concept will perform as anticipated. Currently, the detail design phase is underway. This phase will be followed by initiation of fabrication in late 1972 and ground testing in early 1973. Flight service is planned for mid 1974.

A number of benefits are expected to be achieved with the C-130 composite-reinforced wing program. The design will demonstrate the means for enhancement of structural performance, in this case an improved fatigue life. Fabrication of three full-scale wing boxes will prove the feasibility of manufacturing large



composite-reinforced structures. The flight service program will demonstrate the utility and effectiveness of composites over a long period of time in the real operational environment. The overall benefit is expected to be the longtime flight service experience which will prove the operational capability of composite application at substantial weight savings and will provide confidence for its use in commercial transports.

#### CONCLUDING REMARKS

A review has been presented of the technology development programs that have been undertaken to exploit the concept of composite-reinforced metal structures. These programs have shown that application of the concept provides excellent strength and stiffness improvements to metal structures and also appears to offer a cost effective way to utilize composite materials at the present time. Studies have also indicated that analytical methods presently used for designing metal structures can be modified for the selective reinforcement concept with reasonably accurate results. Although more efficient components should result with better control over residual stresses and better joint design, the present performance of components in ground-based tests indicates significant weight savings over the equivalent all-metal designs. Further validation of the merits of this concept will be obtained when results of planned flight service programs are available. The results are expected to provide much greater confidence in the use of composites for aircraft structures in general and for composite-reinforced metal structures in particular.

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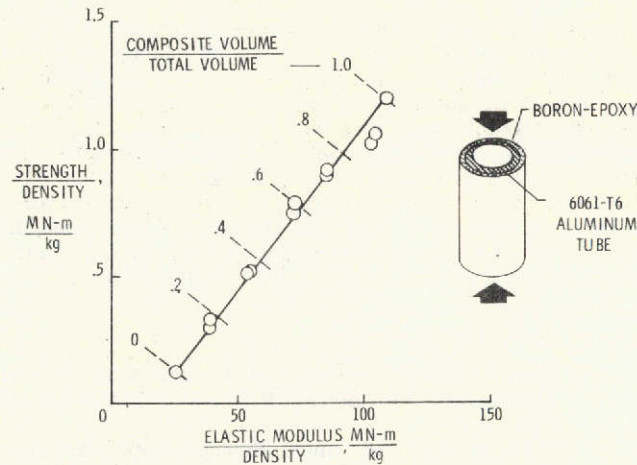


Figure 1. Specific strength and stiffness of 6061-T6 aluminum alloy reinforced with boron-epoxy composite.

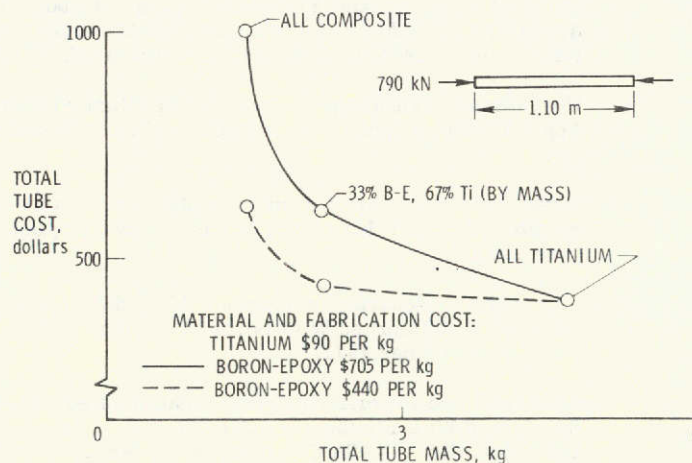


Figure 2. Cost-weight comparison of composite-reinforced tubular column.

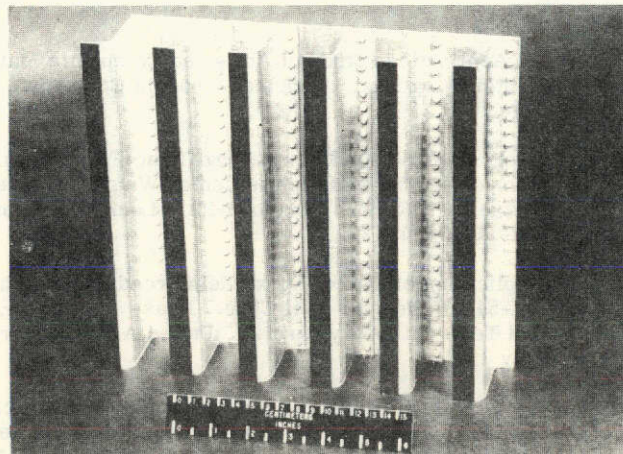


Figure 3. Aluminum skin-stringer panel selectively reinforced with boron-epoxy composite.

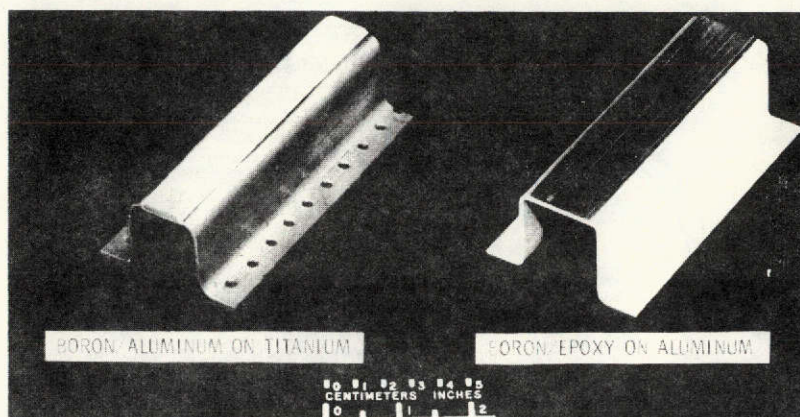


Figure 4. Metal hat stiffeners reinforced with resin-matrix and metal-matrix composites.

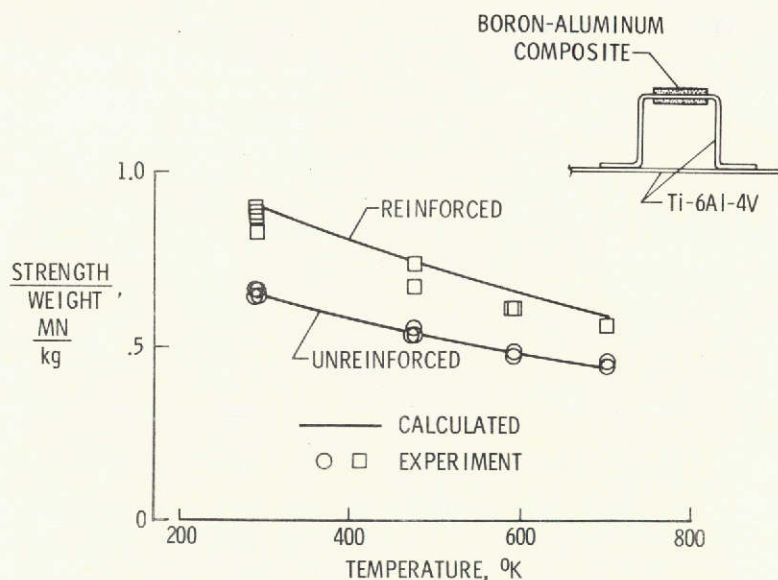


Figure 5. Maximum strength of Ti-6Al-4V panel reinforced with boron-aluminum composite compared with maximum strength of unreinforced titanium panel.

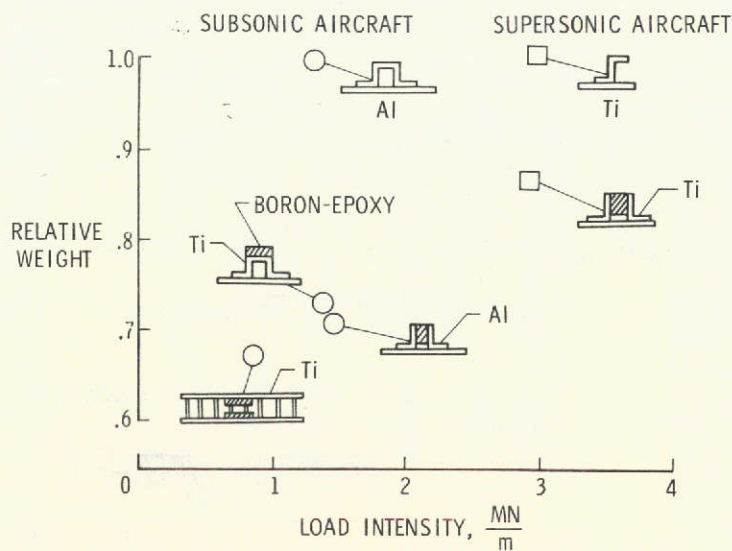


Figure 6. Weight saving potential of aluminum and titanium panels reinforced with boron-epoxy composite.

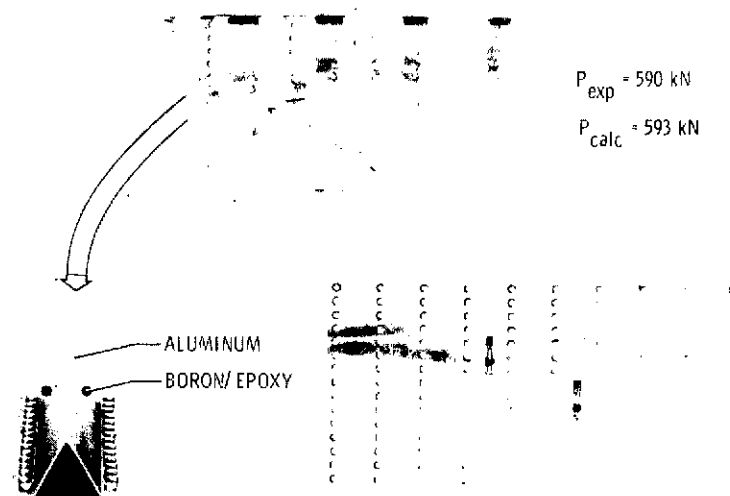


Figure 7. Aluminum panel with Y-section stiffeners containing infiltrated boron-epoxy composite.

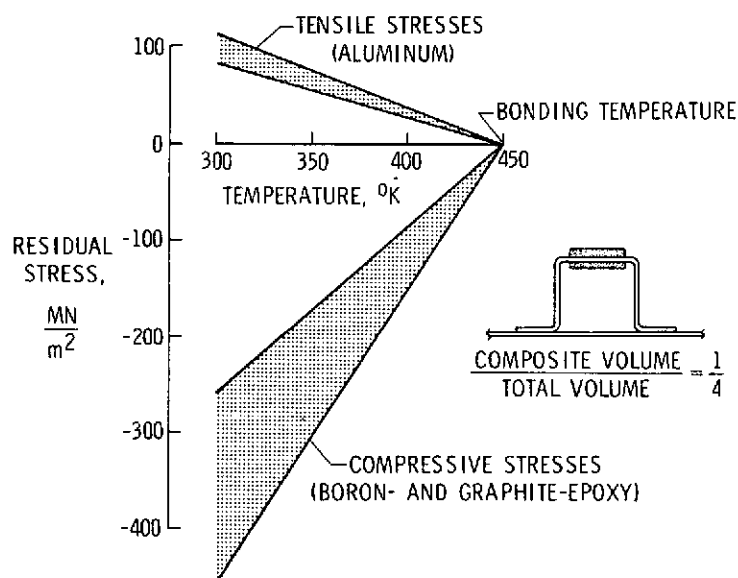


Figure 8. Residual stresses in aluminum components selectively reinforced with boron- and graphite-epoxy composites.

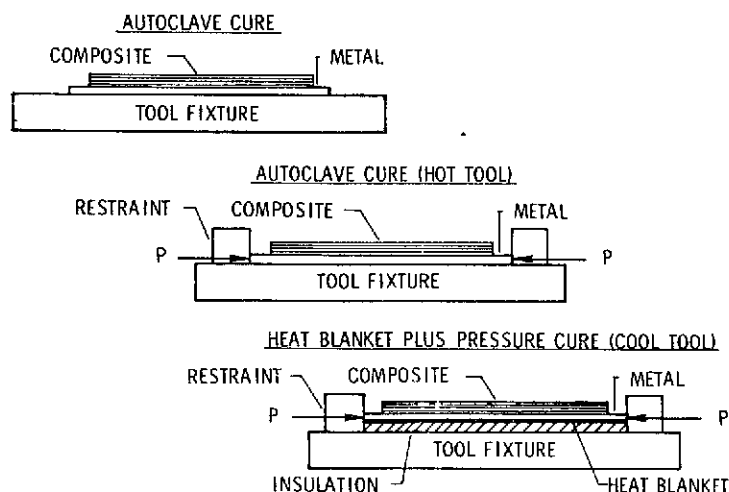


Figure 9. Bonding techniques for unrestrained and restrained metal components selectively reinforced with composites.

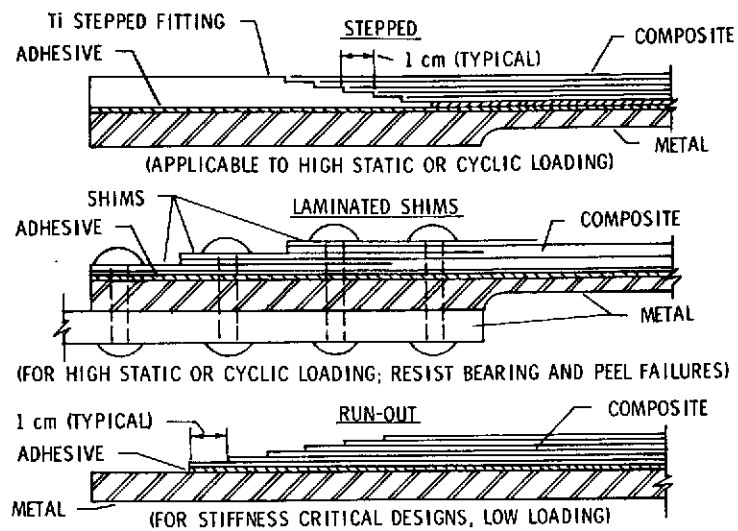


Figure 10. Concepts for composite-to-metal load transfer joints.

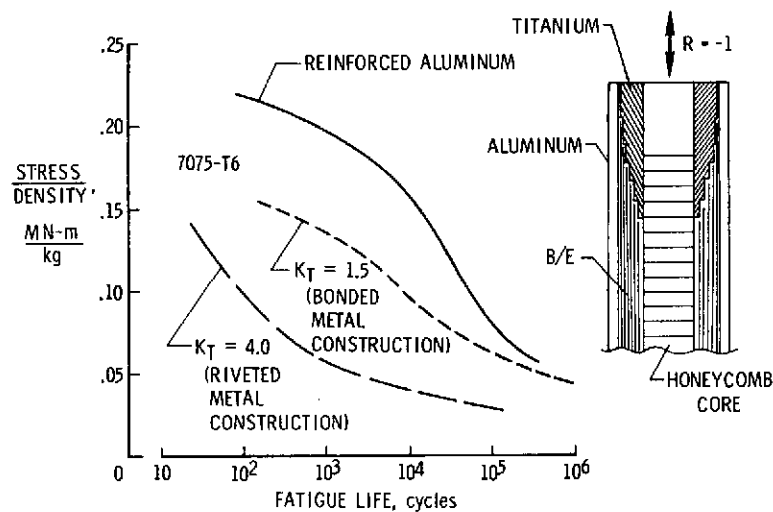


Figure 11. Fatigue life of 7075-T6 aluminum alloy specimen reinforced with boron-epoxy composite for stress ratio,  $R$ , equal to  $-1$ .

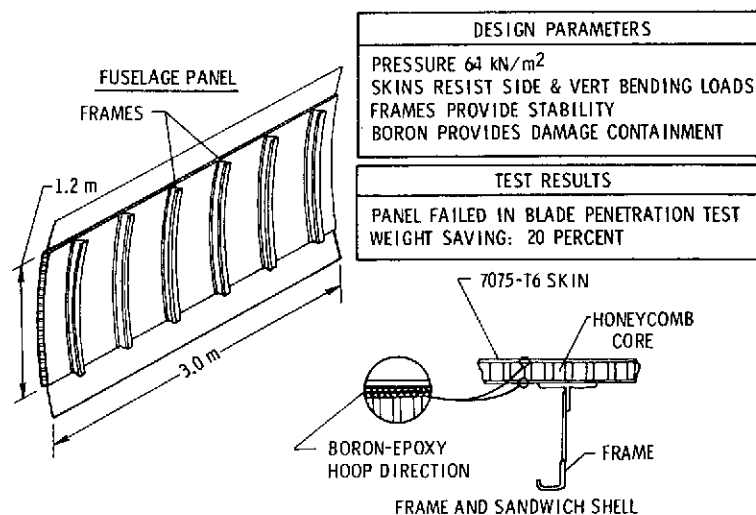


Figure 12. Aluminum fuselage panel reinforced with boron-epoxy composite for tensile (pressurization) test.

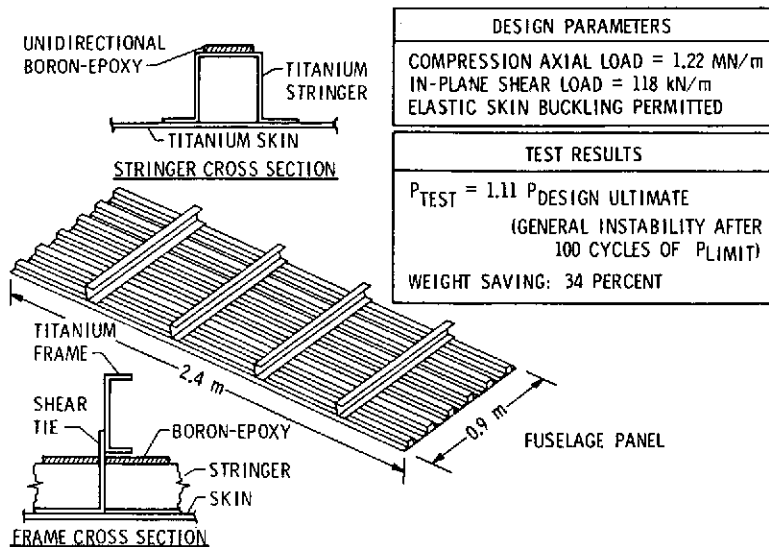


Figure 13. Titanium fuselage panel reinforced with boron-epoxy composite for compression test.

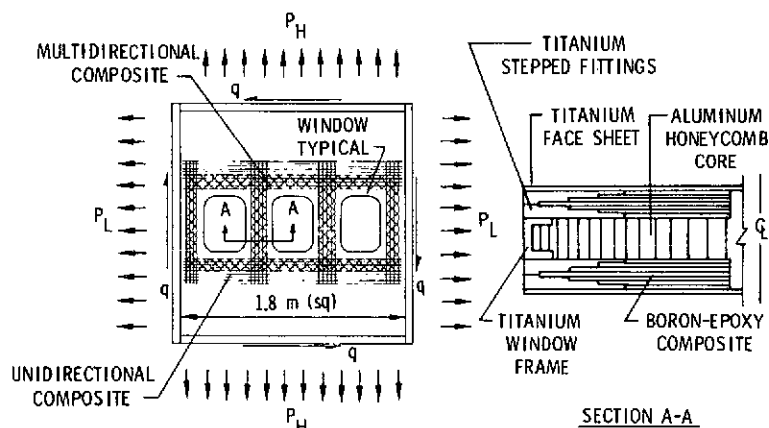


Figure 14. Titanium window-belt panel reinforced with boron-epoxy composite for shear test - configuration.

DESIGN PARAMETERS				TEST RESULTS	
LOAD CASE	$P_H$ , kN/m	$P_L$ , kN/m	$q$ , kN/m	$q_{TEST} = 1.20 q_{DESIGN ULTIMATE}$ WEIGHT SAVING: 25 percent PREDICTED 12 percent ACTUAL	
2.0 $P_i$	420	210	0		
1.5 $P_i + VB$	315	160	265		
1.5 $P_i + SB$	315	590	180		
VB*	0	0	265		

$P_i$  = MAXIMUM INTERNAL PRESSURE  
 VB = VERTICAL BENDING  
 SB = SIDE BENDING (TENSION)  
 VB\* = TEST CONDITION

Figure 15. Titanium window-belt panel reinforced with boron-epoxy composite for shear test - design loads and results.



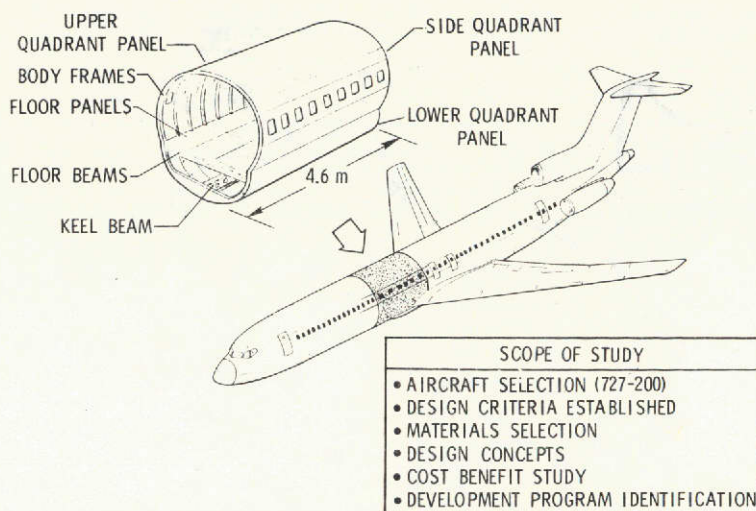
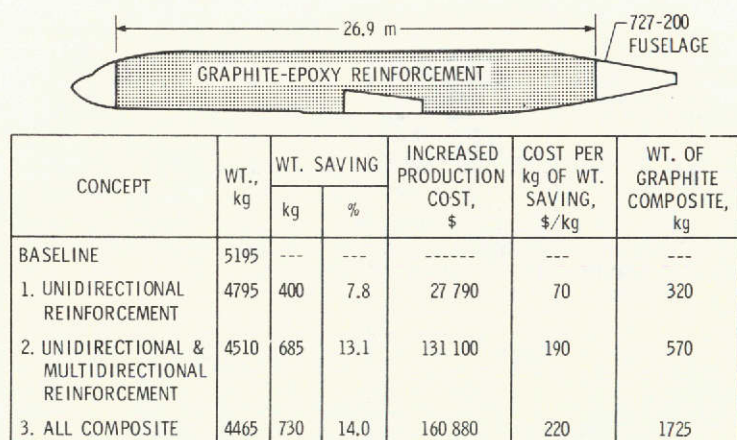


Figure 16. Composites application study for fuselage.



GRAPHITE COMPOSITE COST AT \$130/kg

Figure 17. Weight saving and estimated costs for graphite-epoxy reinforced fuselage.

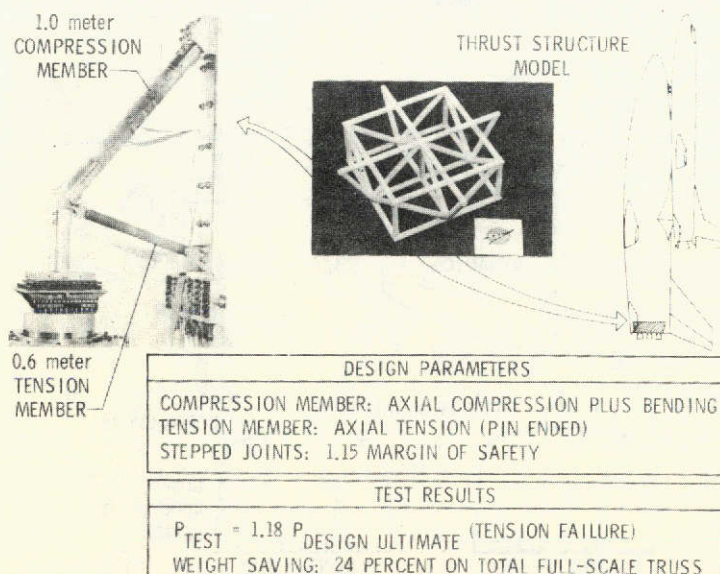


Figure 18. Titanium tubular truss reinforced with boron-epoxy composite for space shuttle booster thrust structure.

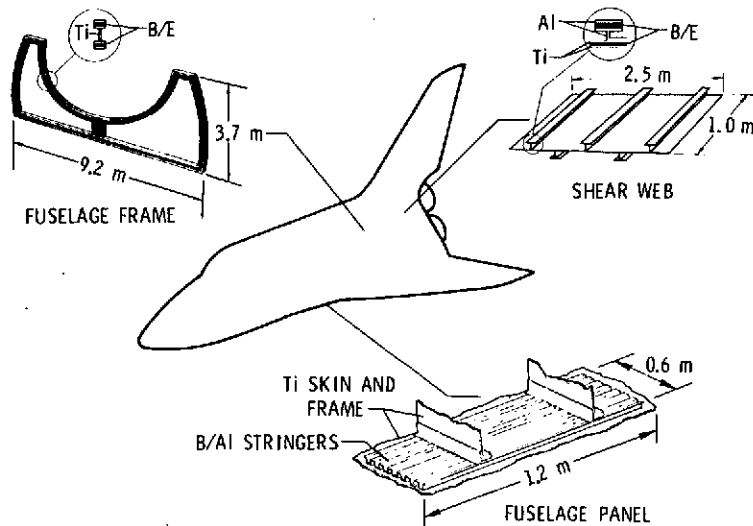


Figure 19. Application of composites to the space shuttle.

COMPONENT	BASE LINE COMPONENT DESIGN	COMPOSITE COMPONENT DESIGN	TEST RESULT	WT. SAVING, %
FRAME	Ti I-BEAM	Ti I-BEAM WITH B/E REINFORCED CAPS	(*)	29
FUSELAGE PANEL	Al SKIN STRINGER	Ti SKIN WITH B/Al HAT STIFFENERS	(*)	46
SHEAR WEB	Ti WEB WITH Al T-SECTION STIFFENERS	B/E WEB CLAD WITH Ti, B/E REINFORCED Al T-SECTION STIFFENERS	(*)	30
LANDING GEAR DOOR	Ti SKINS WITH CHANNEL STIFFENERS	FULL DEPTH Al HC CORE, G/E FACE SHEETS**	$P_{TEST} = 1.6 P_{DESIGN ULT.}$	65
TUBULAR TRUSS	Ti TUBULAR TRUSS	B/E REINFORCED Ti TUBULAR TRUSS	$P_{TEST} = 1.2 P_{DESIGN ULT.}$	24

\* TO BE TESTED.

\*\* ALL COMPOSITE CONSTRUCTION.

Figure 20. Summary of space shuttle component programs.

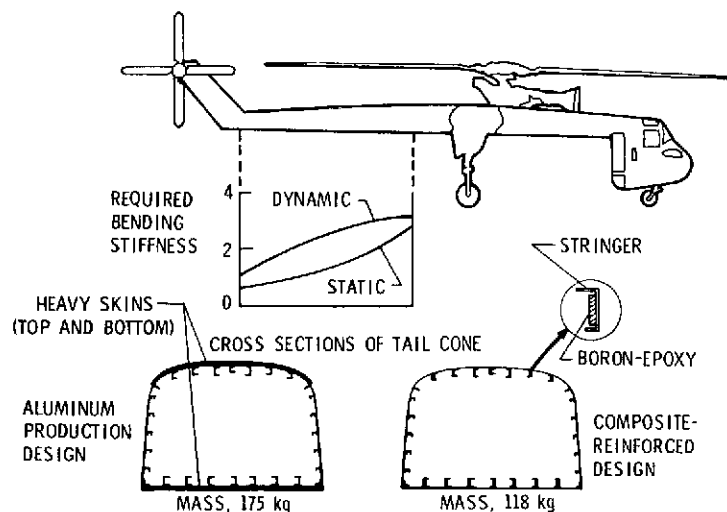


Figure 21. Application of boron-epoxy composite in the aluminum tail cone of the CH-54B helicopter.



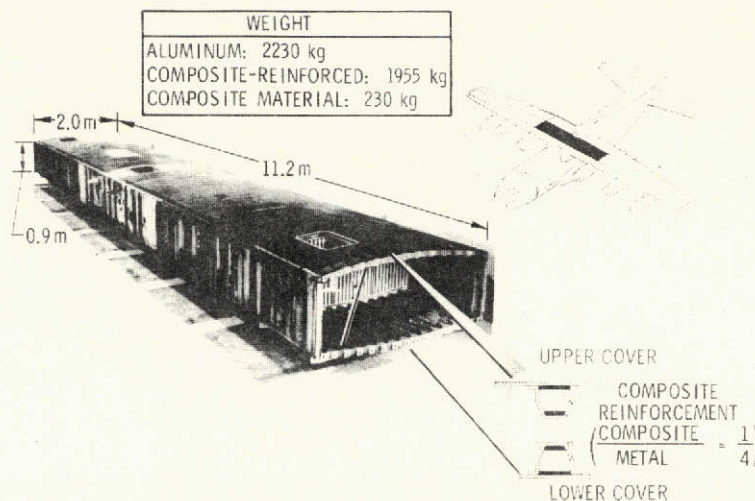


Figure 22. Application of boron-epoxy composite in the aluminum center wing box of the C-130 transport airplane.

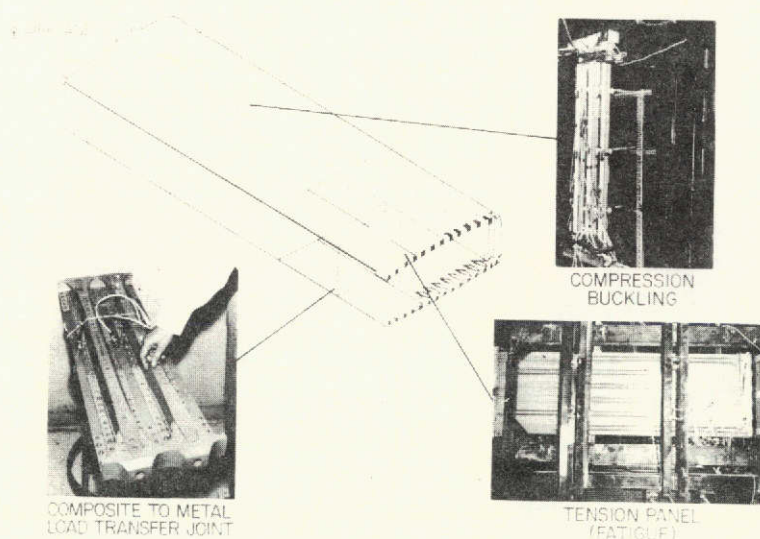


Figure 23. Structural specimens tested in advanced development program on C-130 center wing box.

COMPONENT	TEST RESULTS
COMPRESSION PANEL	$P_{TEST} = 0.96 P_{DESIGN ULTIMATE}$ (END FAILURE RATHER THAN COLUMN BUCKLING)
TENSION PANEL (1)	FATIGUE LIFE > 6 LIFETIMES $P_{RESIDUAL STRENGTH} = 1.09 P_{DESIGN ULTIMATE}$
TENSION PANEL (2)	FATIGUE LIFE > 8 LIFETIMES $P_{RESIDUAL STRENGTH} = 0.92 P_{DESIGN ULTIMATE}$
COMPOSITE-TO-METAL LOAD-TRANSFER JOINT	FATIGUE LIFE > 8 LIFETIMES $P_{RESIDUAL STRENGTH} = 1.35 P_{DESIGN ULTIMATE}$
DESIGN CRITERIA: COMPOSITE-REINFORCED ALUMINUM COMPONENTS TO MEET OR EXCEED STATIC STRENGTH, FATIGUE RESISTANCE, AND DAMAGE CONTAINMENT OF COMPARABLE ALUMINUM COMPONENTS.	

Figure 24. Test results for C-130 wing box reinforced components.